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# RESEARCH MEMORANDUM

CALCULATED PERFORMANCE OF A MERCURY-COMPRESSOR-JET

POWERED AIRPLANE USING A NUCLEAR REACTOR

AS AN ENERGY SOURCE

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CALCULATED PERFORMANCE OF A MERCURY-COMPRESSOR-JET

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SUMMARY

An analysis was made of a system consisting of a mercury turbine-driven air compressor and a mercury condenser wherein heat was added to the compressed air. The heat addition to the mercury is accomplished in an intermediate heat exchanger (mercury boiler) which has a liquid metal, other than mercury, circulating through the opposite side and through a nuclear reactor.

The calculations were made for a flight Mach number of 1.5, an altitude of 45,000 feet, and a turbine-inlet temperature of 1460° R for a range of turbine-inlet pressures, turbine-exhaust pressures, and condenser-inlet Mach numbers. For most of the calculations the lift-drag ratio of the complete airplane was assumed constant at 6.5. For a turbine-inlet pressure of 40 pounds per square inch absolute, a turbine-exhaust pressure of 14 pounds per square inch absolute, and a condenser-inlet Mach number of 0.23, the calculated airplane gross weight required to carry a 20,000 pound payload was 322,000 pounds, the unit volume reactor heat release rate was 8.9 kilowatts per cubic inch, and the maximum reactor wall temperature was approximately 1800° R.

INTRODUCTION

Analytical studies are being made at the NACA Lewis laboratory of various types of propulsion system for aircraft utilizing a nuclear reactor as the energy source. The results of some of these studies on direct-air and binary liquid-metal turbojets are presented in references 1 to 3.

The results of an analysis of a mercury compressor-jet powered airplane using a nuclear reactor as an energy source are presented herein. The system considered consists of an air compressor which is driven by a mercury turbine, and a mercury condenser wherein heat is added to the air. A liquid metal, other than mercury, circulates through

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the reactor and through an intermediate heat exchanger which serves as a mercury boiler. A detailed analytical study of a similar system in which high-pressure water, circulating directly through the reactor, served as the working medium is reported in reference 4.

The results presented herein cover a range of turbine-inlet pressures, turbine-exhaust pressures, and condenser-inlet Mach numbers for a turbine-inlet temperature of  $1460^{\circ}\text{R}$ , a flight Mach number of 1.5, and an altitude of 45,000 feet.

## ANALYSIS

### Description of Powerplant

A schematic diagram of the mercury compressor-jet is shown in figure 1. Air enters the engine through an inlet diffuser and passes through the compressor into the mercury condenser where it is heated by contact with the condenser passage walls. From the condenser the air expands through an exhaust nozzle and discharges at a high velocity to provide jet propulsion. Liquid mercury is pumped into an intermediate (liquid-metal-to-mercury) heat exchanger where it is vaporized and superheated and then expands in a turbine which drives the air compressor. From the turbine the mercury passes through the condenser back to the pump. A liquid metal, other than mercury, circulates through the reactor and intermediate heat exchanger.

### Assumptions

Engine and airplane. - Some of the pertinent assumptions that were made for the engine and airplane for a flight Mach number of 1.5 and an altitude of 45,000 feet are listed in the following table:

Diffuser recovery factor (ratio of actual to theoretical total pressure) . . . . .	0.96
Air-compressor small-stage efficiency . . . . .	0.88
Mercury-turbine adiabatic efficiency . . . . .	0.85
Exhaust-nozzle velocity coefficient . . . . .	0.96
Air-handling capacity of compressor (sea-level static), lb/(sec)(sq ft) of compressor frontal area . . . . .	25
Airplane lift-drag ratio . . . . .	6.5
Ratio of airplane structure to gross weight . . . . .	0.35
Disposable load, lb. . . . .	20,000

For purposes of comparison with the steam compressor-jet of reference 4, the value of lift-drag ratio of the complete airplane was assumed to be 6.5 which is of the same order of magnitude as that used

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in reference 4. In view of the uncertainty in the value of lift-drag ratio for supersonic aircraft, some calculations were made and curves are included for complete airplane lift-drag ratios considerably lower than the value of 6.5. The airplane structure weight fraction which was assumed for this analysis represents a conservative value obtained from a survey of current large aircraft.

Reactor and shield. - The reactor considered in these calculations was assumed to be cylindrical in shape and to have a diameter of 3.5 feet, a length-to-diameter ratio of 0.9, a free-flow area ratio of 0.4, and a weight of 2200 pounds. The reactor was surrounded around the circumference and on both ends by a 3-inch reflector.

The shield was considered to be of the separated type with part of the shielding around the reactor and reflector and part around the crew compartment. The reactor shield consisted of 4 inches of lead around the reactor and reflector and 4 feet of material of specific gravity of 0.85 around the lead resulting in a weight of 84,600 pounds. The crew shield was considered to be a hollow lead cylinder closed on the end facing the reactor and weighing 50,000 pounds.

#### Conditions and Details of Calculations

Calculations were made for a flight Mach number of 1.5 and an altitude of 45,000 feet for a range of turbine-inlet pressures, turbine-exhaust pressures, and condenser-inlet Mach numbers. The calculations for variable turbine-exhaust pressure and variable condenser-inlet Mach number were made holding the other two variables constant. Turbine pressure ratio and condenser-inlet Mach number were held constant for the variable turbine-inlet pressure calculations.

All calculations were made for a turbine-inlet temperature (superheated mercury vapor out of the intermediate exchanger) of 1460° R. The principal reason for choosing this particular temperature was so that a direct comparison at the same temperature level could be made with the steam compressor-jet engine of reference 4.

No values are presented herein for the reactor wall temperatures which would be required to obtain this turbine-inlet temperature of 1460° R. A check calculation using sodium as a reactor coolant indicated that an average reactor wall temperature of about 1600° R and a maximum wall temperature of about 1800° R would be required with the reactor and intermediate heat-exchanger surface areas assumed for this analysis. Some reduction in the reactor wall temperature could be achieved by increasing the size of the intermediate heat exchanger.

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Compressor weights were obtained by extrapolating weight data for compressors which are representative of the lightest of those used in current engines. The weight of the mercury turbine, which represented only a small part of the total powerplant weight, was based on some preliminary turbine-design calculations. Included in the total powerplant weight were such items as pumps, plumbing, reduction gearing, intermediate heat exchanger, and the cycle fluids. For the check calculation of reactor wall temperature previously mentioned the intermediate heat exchanger was assumed to be a counter flow exchanger with the mercury flowing through 0.25-inch-diameter steel tubes.

Condenser weight and performance calculations were based on data on an aircraft fin-and-tube type heat exchanger manufactured by the Harrison Radiator Division of the General Motors Corporation. The weight of this aluminum exchanger, which was originally designed for condensing steam, was recalculated assuming an all steel construction for use in condensing mercury.

Heat-transfer calculations indicated that the over-all heat transfer coefficient for condensing mercury in a steel exchanger would be slightly higher than for condensing steam in an aluminum exchanger. On the basis of these calculations, the manufacturer's steam-to-air heat-dissipation-rate charts for the aluminum exchanger were used for determining mercury-condenser size resulting in what is probably somewhat conservative values for condenser size.

In this analysis it was assumed that the core structure of the condenser was fixed to agree with the core structure of the Harrison fin and tube heat exchanger previously mentioned. This assumption of fixed condenser core dimensions resulted in a definite relation between air compressor pressure ratio and condenser inlet Mach number for given conditions in the liquid circulating loops because the compressor power was fixed by the turbine power. The results of this analysis are plotted against compressor inlet Mach number. A more complete analysis than presented herein in which condenser core configuration is varied would probably reveal a more optimum combination of conditions and hence higher performance than given in the present report. In this respect the results of the present analysis may be considered conservative.

## RESULTS AND DISCUSSION

The performance of the mercury compressor-jet powered airplane is indicated by curves showing the effect of the variables on the airplane gross weight required to carry a 20,000-pound payload, the engine thrust per unit air flow, the engine thrust per unit engine weight (exclusive of reactor and shield weight) and the reactor heat release rate per unit volume. Calculations were made also and curves are presented of the percent liquid in the turbine exhaust.

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Turbine-exhaust pressure. - The effect of turbine-exhaust pressure on airplane gross weight, engine thrust per unit air flow, engine thrust per unit engine weight, reactor heat release rate per unit volume, and percent liquid in the turbine exhaust is shown in figure 2 for a turbine-inlet pressure of 40 pounds per square inch absolute and a condenser-inlet Mach number of 0.23. Figure 2 is plotted at a flight Mach number of 1.5, an altitude of 45,000 feet, a disposable load of 20,000 pounds, and a turbine-inlet temperature of 1460° R.

As the turbine-exhaust pressure decreases, the thrust per unit air flow and the thrust per unit engine weight increase and the airplane gross weight decreases. The increase in thrust per unit air flow with decreasing turbine-exhaust pressure is due principally to an increase in mercury cycle efficiency and, since the engine weight per unit air flow is substantially constant for this range of turbine-exhaust pressures, the thrust per unit engine weight also increases slightly with decreasing turbine-exhaust pressure. For a constant airplane lift-drag ratio and airplane structure weight ratio, the airplane gross weight is a function only of the thrust per unit engine weight and therefore decreases with decreasing turbine-exhaust pressure.

Figure 2 indicates that the minimum turbine-exhaust pressure may be limited by the liquid content in the turbine exhaust, which increases as the turbine-exhaust pressure decreases. Small amounts of liquid in the turbine exhaust probably would not affect the performance of the system materially but large amounts may cause a considerable reduction in the turbine efficiency, which was assumed constant for these calculations. Further reduction in turbine-exhaust pressure below the lowest value investigated (4 lb/sq in. absolute) would eventually result in an increase in airplane gross weight because of a rapidly increasing condenser weight. The condenser weight would increase with decrease in exhaust pressure because more heat-transfer surface would be required on account of the decreasing temperature difference between the entering air and the condensing mercury which in turn is due to the decrease in saturation temperature of the mercury with decreasing pressure.

Turbine-inlet pressure. - The effect of turbine-inlet pressure on airplane gross weight, thrust per unit air flow, thrust per unit engine weight, percent liquid in the turbine exhaust, and reactor unit volume heat release is shown in figure 3 for a turbine pressure ratio of 2.86 and a condenser-inlet Mach number of 0.23. The point at a turbine-inlet pressure of 40 pounds per square inch corresponds to the point at an exhaust pressure of 14 pounds per square inch and an inlet pressure of 40 pounds per square inch in figure 2.

The airplane gross weight and the reactor heat release decrease with increasing turbine-inlet pressure. Because the turbine-pressure ratio is constant, the turbine-exhaust pressure and hence turbine-exhaust temperature (condenser temperature) are increasing as the inlet pressure

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increases. This results in an increase in the temperature of the air leaving the condenser and consequently an increase in the thrust per pound of air flow as shown in figure 3. This increase in thrust per pound of air flow results in a decrease in engine and condenser size and thus accounts for the decrease in airplane gross weight with increasing turbine-inlet pressure.

For constant turbine-pressure ratio, the percent liquid in the turbine exhaust increases with increasing turbine-inlet pressure as shown and would eventually seriously affect the system performance. Plots similar to those shown in figure 3 except for a constant turbine-exhaust pressure (rather than constant pressure ratio) would show a more rapid increase in liquid content with increasing turbine-inlet pressure.

Condenser-inlet Mach number. - The effect of condenser-inlet Mach number on airplane gross weight, thrust per unit air flow, thrust per unit engine weight and reactor heat release per unit volume are shown in figure 4 for a turbine-inlet pressure of 40 pounds per square inch absolute and a turbine-exhaust pressure of 14 pounds per square inch absolute. The airplane gross weight required to carry a 20,000-pound pay load has a minimum value of 318,000 pounds at a condenser-inlet Mach number of about 0.20. The reactor heat release has a minimum value of 7.5 kilowatts per cubic inch at a condenser-inlet Mach number of about 0.13.

As the condenser-inlet Mach number is increased above the lowest value shown in figure 4 the condenser air-handling capacity increases with a consequent decrease in condenser size and airplane gross weight. At the same time, however, the air-side pressure drop in the condenser is increasing, causing the thrust per unit air flow to decrease as shown. This decrease in thrust per unit air flow results in an increase in engine size (in order to maintain the thrust) and eventually causes the airplane gross weight to increase with increasing condenser-inlet Mach number.

Although the effect of turbine-inlet temperature was not investigated in this analysis, it might be mentioned that an increase in turbine-inlet temperature would result in an initial decrease in airplane gross weight for constant inlet and exhaust pressures. For constant inlet and exhaust pressures, increasing the turbine-inlet temperature beyond a certain point results in superheated vapor in the turbine exhaust, which must be cooled before it can be condensed. The heat-transfer coefficients for cooling a superheated vapor are generally much lower than for condensing vapor so that the exchanger surface areas required for this cooling will be considerably larger than those required for condensing. The presence of the superheated vapor in the turbine exhaust, however, has the advantage of resulting in higher air temperature rises in the condenser and hence higher values of thrust per unit air flow. Along

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with the increase in turbine-inlet temperature there is always, of course, the disadvantage of higher reactor wall temperatures. Further analysis of the inlet-temperature effect and the effect of the superheated vapor in the turbine exhaust at various combinations of turbine inlet and exhaust pressures is desirable.

Effect of nacelle drag. - In the previous figures, a value for the lift-drag ratio of the complete airplane of 6.5 was used to permit comparison with the results of reference 4 in which a "split-wing" configuration with the engines between the wings was assumed. In view of the uncertainty of the actual lift-drag ratios obtainable with the split-wing configurations, additional calculations were made for two more conventional type installations, one a flying-wing type having only wing and fuselage drag, and the other a wing-fuselage type having wing, fuselage, nacelle, and tail surface drag. In both cases the engines are contained in nacelles attached to the wing. The results of these calculations are presented in figure 5 where, as in the previous figure, airplane gross weight is plotted against condenser-inlet Mach number. The curve for the constant airplane lift-drag ratio from figure 4 is included for comparison.

For the drag calculations, the condenser was assumed to be inclined with the inlet face at approximately  $30^\circ$  to the horizontal and the maximum nacelle frontal area was assumed to be 5 percent larger than the compressor frontal area or one-half the condenser frontal area, whichever was greatest. The tail drag was assumed to be 15 percent of the wing drag and the fuselage was assumed to be 7 feet in diameter and 140 feet long. A profile drag coefficient of 0.2 (based on frontal area) and a friction drag coefficient of 0.003 were assumed for both the nacelle and the fuselage. The lift-drag ratio of the wing was assumed to be 13, a value which is felt to be about the maximum attainable with sweptback wings.

The trend of airplane gross weight with condenser-inlet Mach number is essentially the same for all three configurations. For cases II and III, (fig. 5) however, the gross weight increases much more rapidly as the condenser-inlet Mach number is changed from the optimum value.

The following table gives the minimum airplane gross weights from figure 5 for the three cases along with the corresponding reactor heat-release rates and complete airplane lift-drag ratios.



Case	Airplane gross weight, (lb)	Complete airplane lift-drag ratio	Reactor heat-release rate per unit volume (kw/cu in.)
I	318,000	6.5	7.4
II	403,000	4.0	17.0
III	538,000	2.9	31.3

The airplane lift-drag ratio for the flying wing is about 60 percent of the value assumed for the split-wing configuration. This value results in a 26-percent increase in gross weight and more than doubles the reactor heat-release rate. The wing-fuselage configuration has a gross weight about 70 percent higher than the split-wing configuration and the reactor heat-release rate is higher by a factor of over 4.

Experimental data on which to base calculations of the drag of the various configurations at supersonic speeds is meager. The large effect of the drag of the configuration on the gross weight of the airplane and the propulsive power required indicates a need for experimental aerodynamic studies of configurations suitable for nuclear propulsion at supersonic velocities.

General. - The following table presents a gross weight breakdown along with some pertinent engine and reactor variables for a representative operating condition:

Flight Mach number . . . . .	1.5
Altitude, ft . . . . .	45,000
Airplane lift-drag ratio . . . . .	6.5
Turbine-inlet temperature, °R. . . . .	1460
Turbine-inlet pressure, lb/sq inch absolute. . . . .	40
Turbine-exhaust pressure, lb/sq inch absolute. . . . .	14
Condenser-inlet Mach number. . . . .	0.23
Reactor core diameter, ft. . . . .	3.5
Reactor heat release, kw/cu in. . . . .	8.9
Compressor-pressure ratio. . . . .	1.17
Compressor frontal area, sq ft . . . . .	379
Condenser frontal area, sq ft. . . . .	579
Engine thrust, lb . . . . .	49,500
System weight lb:	
Reactor . . . . .	2000
Reactor shield . . . . .	85,000
Crew shield. . . . .	50,000

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Engine weight, lb	
Compressor and drive shaft. . . . .	10,700
Turbine, reduction gear, feed pump and plumbing . . . . .	7000
Intermediate exchanger . . . . .	4600
Working fluids . . . . .	4700
Condenser weight. . . . .	26,000
Total engine weight . . . . .	53,000
Airplane structure weight, lb . . . . .	112,000
Payload, lb . . . . .	20,000
Airplane gross weight, lb . . . . .	322,000

An accurate determination of the engine weight requires the making of a complete engine layout and design study of the components. This was not done and hence the engine weights used in this analysis are approximate. They are believed to be sufficiently accurate, however, to give the proper order of magnitude of the airplane gross weight and trends. A change in engine weight (exclusive of the condenser) of 20 per cent, for example, causes a change of between 2 and 4 percent in airplane weight for the range of conditions covered in this report and hence an error of this magnitude would not affect conclusions regarding the feasibility of this engine.

The condenser may be installed in the airplane in such a way that all or part of the total condenser frontal area may be submerged so that the airplane need not necessarily be charged with the additional frontal area. If the condenser is installed in a nacelle with the other engine components or in a separate nacelle it may be inclined to the line of flight with a considerable reduction in frontal area over the value of 579 square feet shown in the previous table.

With the condenser configuration used in this analysis the compressor-pressure ratios for the system were between 1.13 and 1.30 for all the conditions investigated.

The analytical study of reference 4 indicates that a nuclear steam-compressor-jet powered airplane, similar to the mercury system considered herein, designed to fly at a flight Mach number of 1.5 and an altitude of 45,000 feet with a bomb load of 20,000 pounds would have a gross weight of 236,000 pounds. The unit volume reactor heat-release rate at this flight condition would be about 5 kilowatts per cubic inch. The shield weight and the ratio of airplane structure weight to gross weight which were used in the analysis of reference 4 were considerably lower than those used in this analysis. With comparable assumptions as to airplane structure weight and shielding requirements, the airplane gross weights and the reactor heat-release rates per unit volume for the steam and mercury systems would be about the same.

The mercury system has the disadvantage of requiring a liquid-metal cooled reactor, which for its practical realization presents a multitude

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of design and development problems. Also, the mercury system with its extra heat exchanger and two fluid cycle may require somewhat higher reactor surface temperatures than the steam system for a given turbine-inlet temperature.

The steam compressor-jet system reported in reference 4 has the disadvantage of operating at reactor and turbine-inlet pressures of 5000 pounds per square inch and a condenser pressure of 680 pounds per square inch absolute. For the same performance, the mercury system can operate at much lower pressures (for example, 40 lb/sq in. absolute turbine-inlet pressure and 14 lb/sq in. absolute condenser pressure) and thus avoids the many complications associated with the design of a high-pressure reactor, turbine, and condenser.

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### SUMMARY OF RESULTS

The results of calculations on the performance of a mercury compressor-jet powered airplane using a nuclear reactor as the energy source may be summarized as follows:

1. For constant turbine-inlet temperature and payload, the airplane gross weight and the reactor heat release per unit volume decreased with decreasing turbine-exhaust pressure (turbine-inlet pressure constant) and increasing turbine-inlet pressure (turbine pressure ratio constant). The liquid content in the turbine exhaust increased with decreasing turbine-exhaust pressure and increasing turbine-inlet pressure and will limit the useful minimum exhaust and maximum inlet turbine pressures.

2. The airplane gross weight and the reactor heat release rate per unit volume had minimum values for the conditions considered at condenser-inlet Mach numbers of about 0.20 and 0.13, respectively.

3. For a flight Mach number of 1.5, an altitude of 45,000 feet, an airplane lift-drag ratio of 6.5, a turbine-inlet temperature of 1460° R, a turbine-inlet pressure of 40 pounds per square inch absolute, a turbine-exhaust pressure of 14 pounds per square inch absolute, a condenser-inlet Mach number of 0.23, a condenser total pressure ratio of 1.15, a reactor diameter of 3.5 feet, and assuming a divided-type shield, the calculated airplane gross weight required to carry a 20,000 pound payload was 322,000 pounds, the reactor heat release per unit volume was 8.9 kilowatts per cubic inch and the maximum reactor wall temperature was about 1800° R. These do not represent optimum design conditions.

4. For most of the calculations, the lift-drag ratio of the complete airplane was assumed constant at a value of 6.5. A few calculations were made, however, assuming the lift-drag ratio of the wing constant at a value of 13 and calculating the drags of the various components separately.

Two configurations considered were (a) a flying wing having only wing and fuselage drag, and (b) a more conventional wing-fuselage configuration having wing, fuselage, nacelle, and tail surface drag. For the same engine and flight conditions as tabulated in item 3, the flying-wing airplane had a gross weight of 403,000 pounds and the reactor heat release per unit volume was 17.0 kilowatts per cubic inch. The wing-fuselage configuration for the same conditions had a gross weight of 538,000 pounds and the reactor heat release was 31.3 kilowatts per cubic inch.

Lewis Flight Propulsion Laboratory,  
National Advisory Committee for Aeronautics,  
Cleveland, Ohio, April 5, 1951.

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3. Doyle, R. B.: Calculated Performance of a Direct-Air Nuclear Turbojet-Powered Airplane Using a Split-Flow Reactor and a Separated-Type Shield. NACA RM E50K06, 1950.
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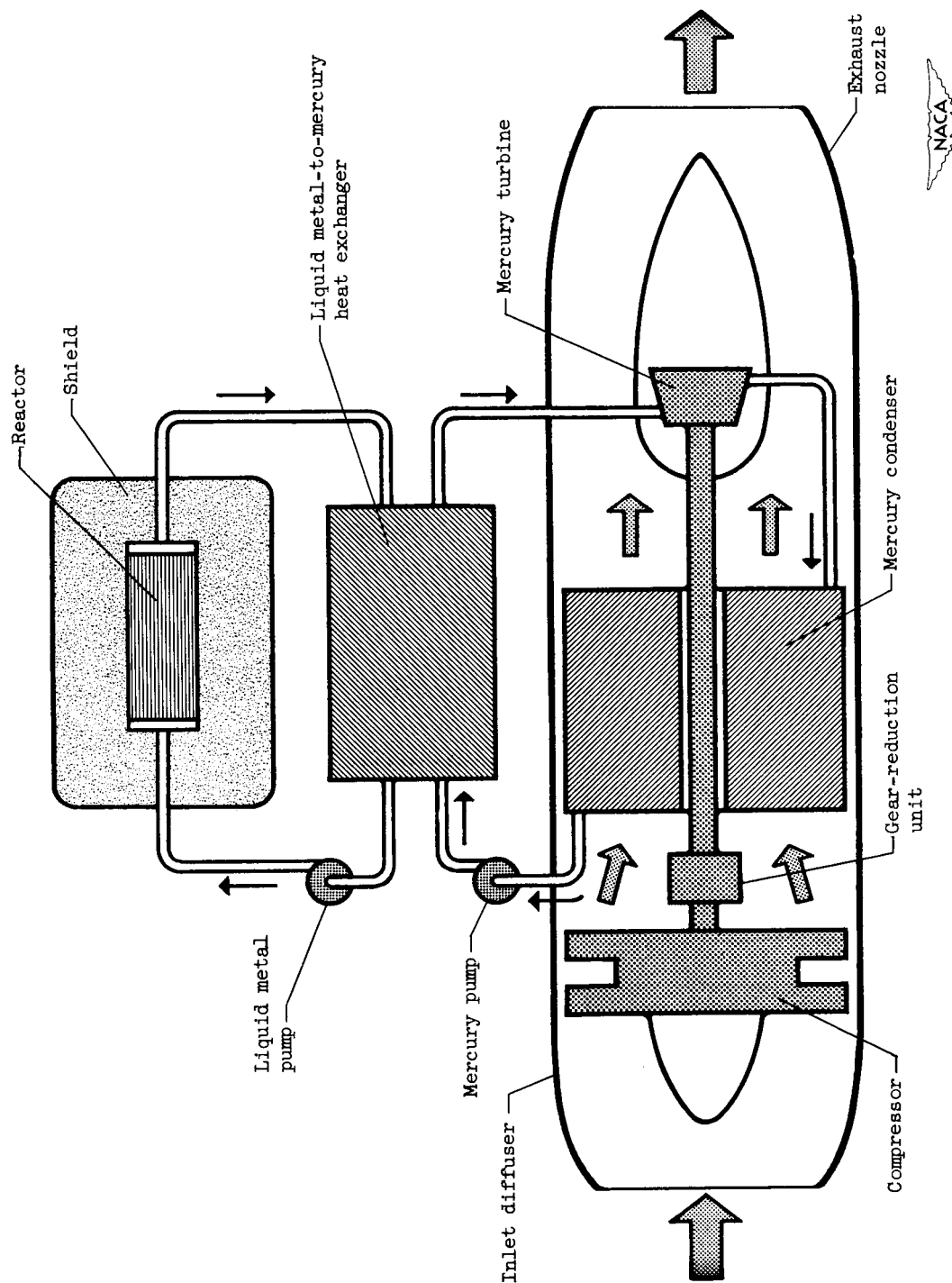


Figure 1. - Schematic diagram of mercury compressor jet.

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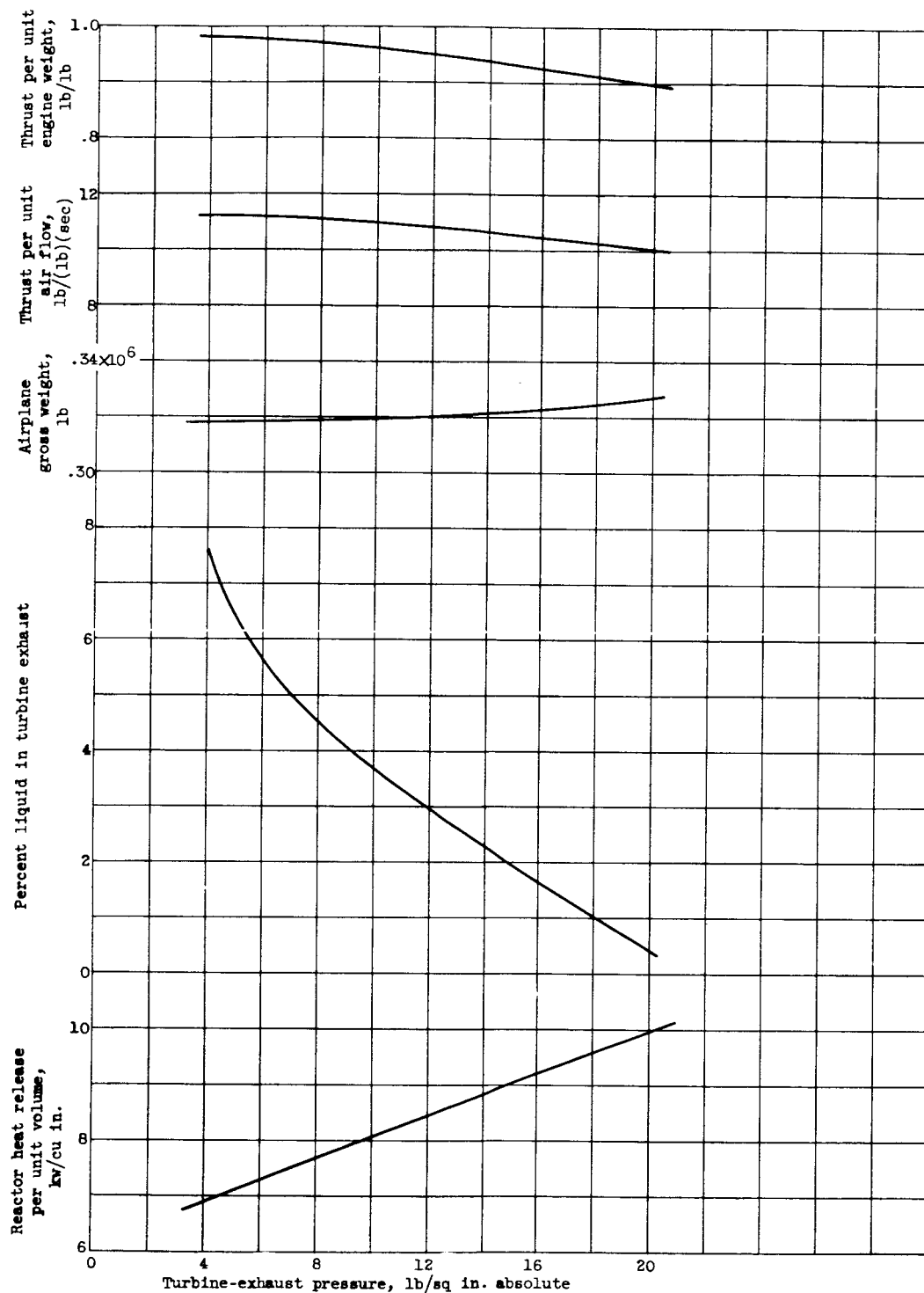


Figure 2. - Variation of airplane gross weight, thrust per unit air flow, thrust per unit engine weight, reactor heat release per unit volume, and percent liquid in turbine exhaust with turbine-exhaust pressure. Flight Mach number, 1.5; altitude, 45,000 feet; disposable load, 20,000 pounds; turbine-inlet temperature, 1460° R; turbine-inlet pressure; 40 pounds per square inch absolute; approximate condenser-inlet Mach number, 0.23.

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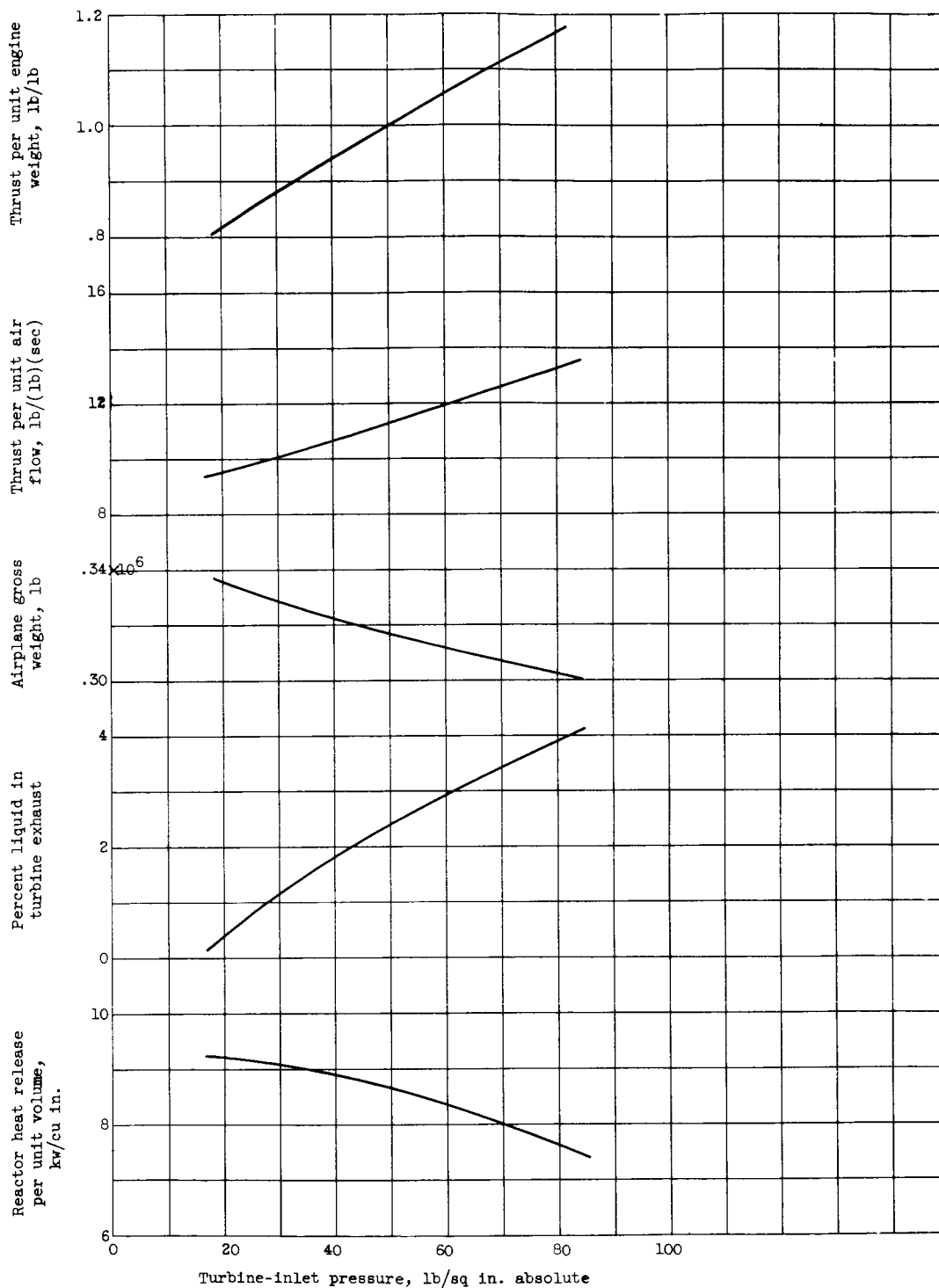


Figure 3. - Variation of airplane gross weight, thrust per unit air flow, thrust per unit engine weight, reactor heat release per unit volume, and percent liquid in turbine exhaust with turbine-inlet pressure. Flight Mach number, 1.5; altitude, 45,000 feet; disposable load, 20,000 pounds; turbine-inlet temperature, 1460° R; turbine pressure ratio, 2.86; approximate condenser-inlet Mach number, 0.23.

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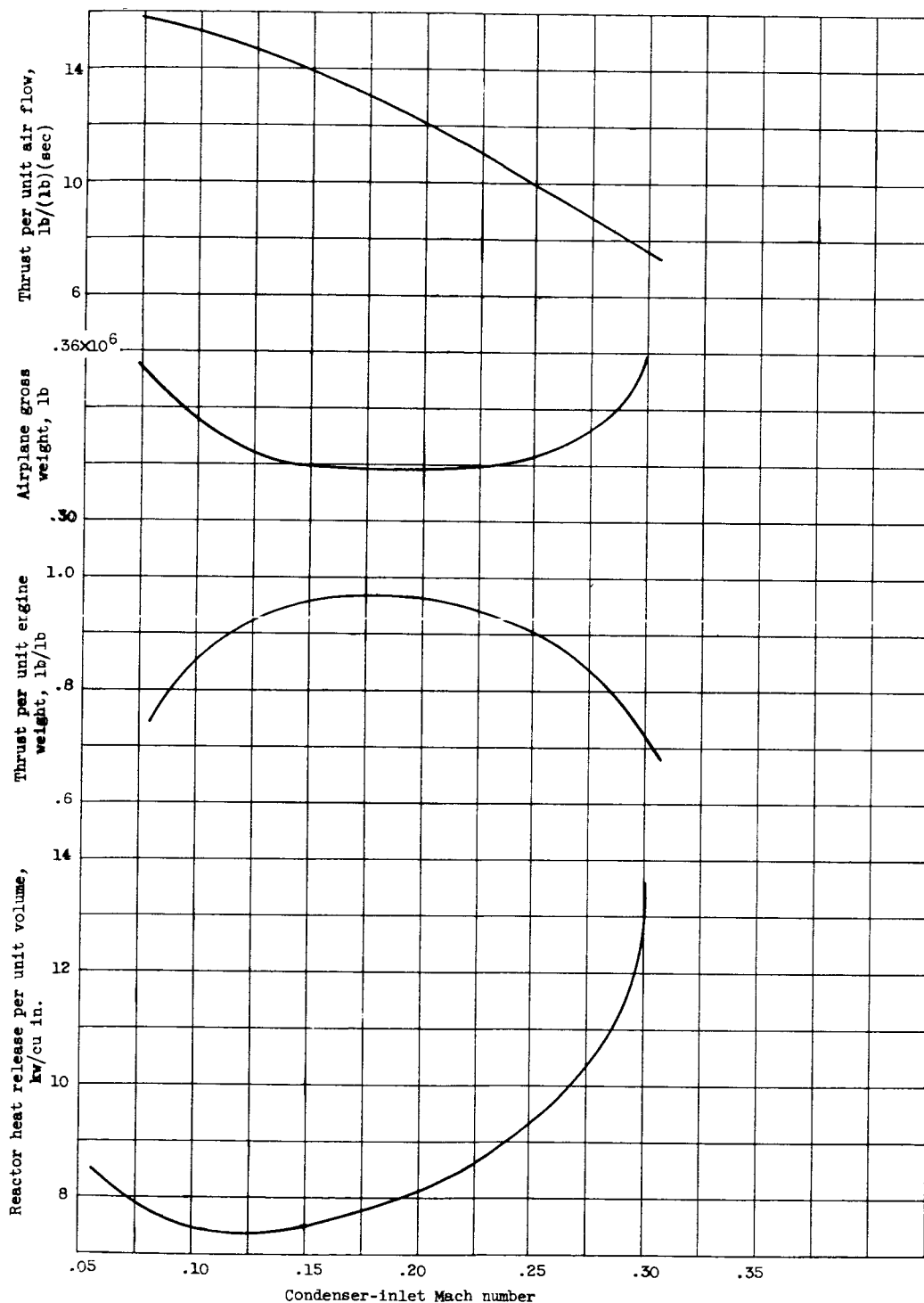


Figure 4. - Variation of airplane gross weight, thrust per unit air flow reactor heat release per unit volume, and thrust per unit engine weight with condenser-inlet Mach number. Flight Mach number, 1.5; altitude, 45,000 feet; disposable load, 20,000 pounds; turbine-inlet temperature,  $1460^{\circ}$  R; turbine-inlet pressure, 40 pounds per square inch absolute; turbine-exhaust pressure, 14 pounds per square inch absolute.



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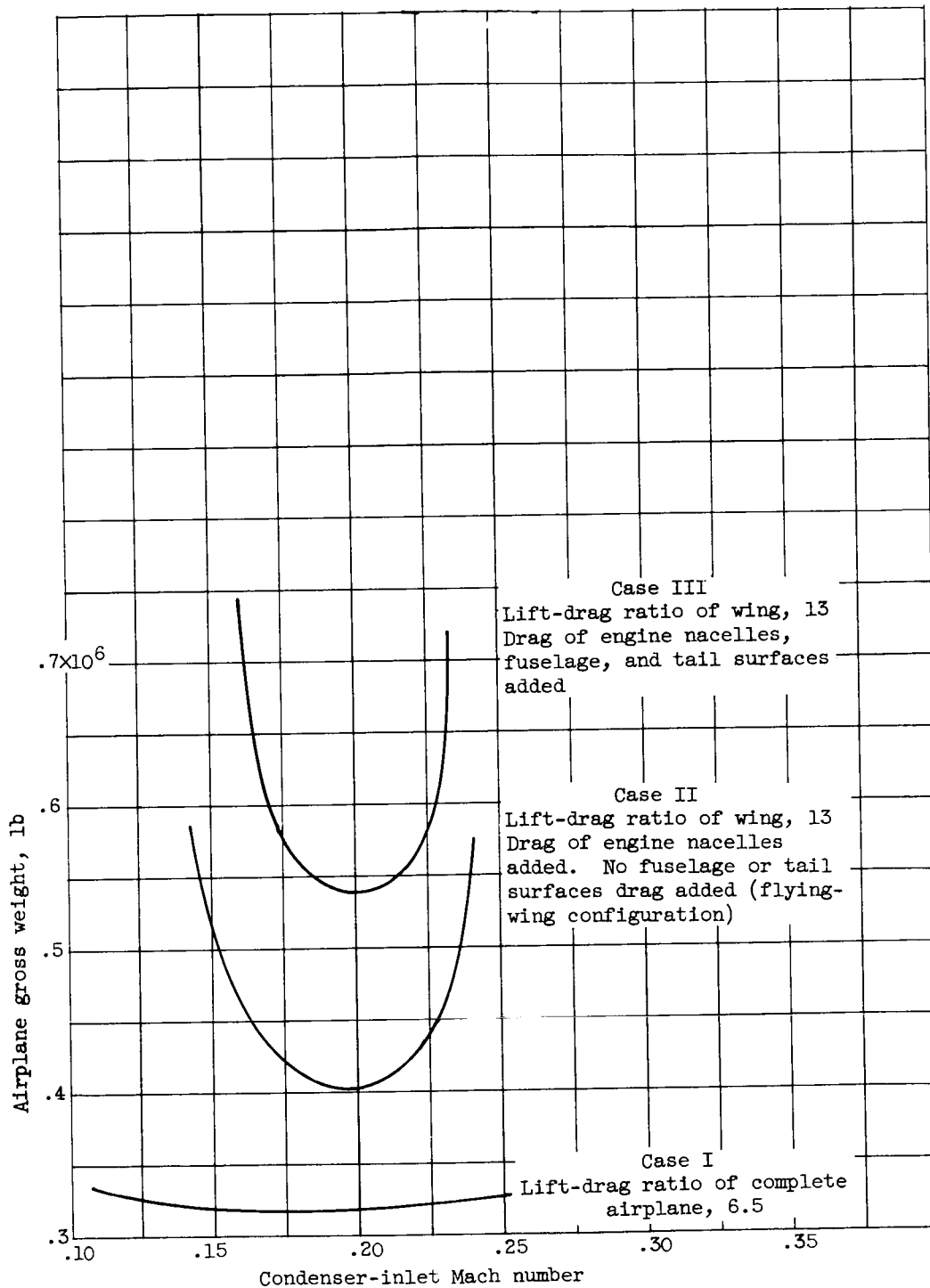


Figure 5. - Variation of airplane gross weight with condenser-inlet Mach number for various lift-drag ratio assumptions. Flight Mach number, 1.5; altitude, 45,000 feet; disposable load, 20,000 pounds; turbine-inlet temperature, 1460° R; turbine-inlet pressure, 40 pounds per square inch absolute; turbine-exhaust pressure, 14 pounds per square inch absolute.

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